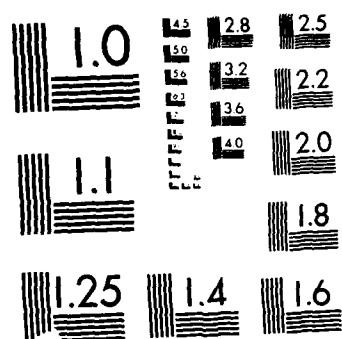AD-A207 584   HYBRID ELECTRIC CHEMICAL PROPULSION(U) AEROSPACE CORP   1/1
               EL SEGUNDO CA AEROPHYSICS LAB   J E POLLARD ET AL.
               01 APR 89 TR-0086(6930-03)-1 SD-TR-89-24
UNCLASSIFIED   F04701-85-C-0086   F/G 21/8.1   NL

1.0 45 2.8 2.5
50
56 3.2 2.2
63 3.6
40 2.0

1.1

1.25 1.4 1.6

1.8

UTION TEST CHART

AD-A207 584

Hybrid Electric Chemical Propulsion

J. E. POLLARD and R. B. COHEN
Aerophysics Laboratory
Laboratory Operations
The Aerospace Corporation
El Segundo, CA 90245
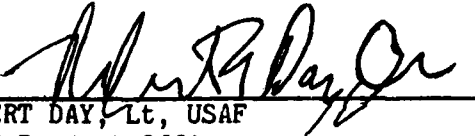
1 April 1989

DTIC
ELECTE
MAY 11 1989
S
H
D

89 5 11 129

This report was submitted by The Aerospace Corporation, El Segundo, CA 90245, under Contract No. F04701-85-C-0086 with the Space Division, P.O. Box 92960, Worldway Postal Center, Los Angeles, CA 90009. It was reviewed and approved for The Aerospace Corporation by W. P. Thompson, Jr., Director, Aerophysics Laboratory.

Lt Robert Day, SD/CLUE, was the project officer for the Mission-Oriented Investigation and Experimentation (MOIE) Program.

This report has been reviewed by the Public Affairs Office (PAS) and is releasable to the National Technical Information Service (NTIS). At NTIS, it will be available to the general public, including foreign nationals.

This technical report has been reviewed and is approved for publication. Publication of this report does not constitute Air Force approval of the report's findings or conclusions. It is published only for the exchange and stimulation of ideas.

ROBERT DAY, Lt, USAF
MOIE Project Officer
SD/CLUE

JAMES A. BERES, Lt Col, USAF
Director, AFSTC West Coast Office
AFSTC/WCO

REPORT DOCUMENTATION PAGE

1a. REPORT SECURITY CLASSIFICATION	1b. RESTRICTIVE MARKINGS
Unclassified	

2a. SECURITY CLASSIFICATION AUTHORITY	3. DISTRIBUTION/AVAILABILITY OF REPORT
	Approved for public release;
2b. DECLASSIFICATION/DOWNGRADING SCHEDULE	distribution unlimited.

4. PERFORMING ORGANIZATION REPORT NUMBER(S)	5. MONITORING ORGANIZATION REPORT NUMBER(S)
SD-TR-89-24	TR-0086(6930-03)-1

6a. NAME OF PERFORMING ORGANIZATION	6b. OFFICE SYMBOL (If applicable)	7a. NAME OF MONITORING ORGANIZATION
The Aerospace Corporation Laboratory Operations		Space Systems Division

6c. ADDRESS (City, State, and ZIP Code)	7b. ADDRESS (City, State, and ZIP Code)
El Segundo, CA 90245	Los Angeles Air Force Base Los Angeles, CA 90009-2960

8a. NAME OF FUNDING/SPONSORING ORGANIZATION	8b. OFFICE SYMBOL (If applicable)	9. PROCUREMENT INSTRUMENT IDENTIFICATION NUMBER
		F04701-85-C-0086

8c. ADDRESS (City, State, and ZIP Code)	10. SOURCE OF FUNDING NUMBERS			
	PROGRAM ELEMENT NO.	PROJECT NO.	TASK NO.	WORK UNIT ACCESSION NO.

11. TITLE (Include Security Classification)
Hybrid Electric Chemical Propulsion

12. PERSONAL AUTHOR(S)
Pollard, James E., and Cohen, Ronald B.

13a. TYPE OF REPORT	13b. TIME COVERED FROM ___ TO ___	14. DATE OF REPORT (Year, Month, Day) 1989 April 1	15. PAGE COUNT 26

16. SUPPLEMENTARY NOTATION

17. COSATI CODES			18. SUBJECT TERMS (Continue on reverse if necessary and identify by block number)
FIELD	GROUP	SUB-GROUP	Auxiliary propulsion, Electrically augmented H_2/O_2 thrusters
			Chemically augmented electrothermal thrusters, Electrothermal thrusters
			Electric propulsion, Spacecraft propulsion.

19. ABSTRACT (Continue on reverse if necessary and identify by block number)

A spacecraft propulsion concept that uses a combination of electrical energy and chemical energy for thrust is explored using thermodynamic modeling calculations. The essence of this concept consists of adding a carefully chosen amount of O_2 or F_2 oxidizer to the propellant flow of a conventional H_2 electrothermal thruster. A general method is given for selecting the fuel:oxidizer ratio so as to optimize the thruster performance for any set of mission constraints.

20. DISTRIBUTION/AVAILABILITY OF ABSTRACT	21. ABSTRACT SECURITY CLASSIFICATION
☑ UNCLASSIFIED/UNLIMITED ☐ SAME AS RPT. ☐ DTIC USERS	Unclassified

22a. NAME OF RESPONSIBLE INDIVIDUAL	22b. TELEPHONE (Include Area Code)	22c. OFFICE SYMBOL

DD FORM 1473, 84 MAR
83 APR edition may be used until exhausted.
All other editions are obsolete.

SECURITY CLASSIFICATION OF THIS PAGE

PREFACE

We wish to thank Karen L. Foster for assistance with the computations.

CONTENTS

FIGURES

I. INTRODUCTION

Electrothermal propulsion systems operating in the specific impulse range from 300 s to 2000 s are likely to find application in such near-term missions as drag make-up and attitude control for space stations[1,2] and in certain types of orbital transfers.[3,4] Electrothermal devices are attractive because (1) they are relatively simple to construct and operate, (2) they can be used with noncontaminating propellants, and (3) their I_{sp} range is appropriate for the modest electrical power levels (below 100 kW) that will be available during the next decade. Electrically augmented hydrazine resistojets (I_{sp} = 300 s, P $\cong$ 0.5 kW) are already performing station-keeping functions for a number of geosynchronous communications satellites,[2] representing the first step in the transition to the widespread use of electric propulsion.

A hybrid electric chemical propulsion concept is proposed here that is potentially the next logical step beyond the hydrazine resistojet and that spans the I_{sp} range between high-performance liquid rockets (450 s) and hydrogen electrothermal arcjets (~1000 s). The essence of this concept is to deposit electrical energy into a pair of reactive propellants (H_2/O_2 and H_2/F_2 are considered here) and to choose the fuel:oxidizer ratio so as to make an optimized trade-off between the amount of electrical energy and the amount of chemical energy being consumed. This choice depends on the size of the electrical power source and on the mass of propellant available for a given maneuver. A reduction in the initial vehicle mass at the expense of a longer trip time is achieved if relatively less oxidizer is used, reaching in the upper I_{sp} limit the case of pure hydrogen electrothermal propulsion. A reduction in the trip time and a corresponding increase in the initial mass is achieved if relatively more oxidizer is used, reaching in the lower I_{sp} limit the case of bipropellant chemical propulsion. This trade-off between mass flow (or thrust) and I_{sp} is common to every electric propulsion scheme. However, the new facet that our analysis emphasizes is that chemical energy can be added in a precise manner to improve the I_{sp} vs. thrust trade-off beyond what is achievable using purely electrical means.

5

The concept of a thermally augmented H_2/O_2 rocket engine recently was given a careful analysis by Frisbee[5] from a perspective somewhat different from that in the present work. The approach here is perhaps a more general treatment of the problem than has been given previously and might be distinguished by being called a "chemically augmented electrothermal" propulsion scheme.

II. MODEL AND ASSUMPTIONS

A thermodynamic model of hybrid propulsion is used to calculate I_{sp} as a function of the electrical energy deposited per unit mass of propellant, $\Delta E/m$.

The assumptions made here are that (1) there is adiabatic, constant pressure combustion; (2) thermochemical equilibrium is achieved upstream of the nozzle; (3) propellants are initially at room temperature, 298 K; and (4) there is isentropic nozzle expansion to infinite Mach number with no further chemical reactions in the nozzle. Some of the more important approximations inherent in these assumptions are, respectively, (1) neglect of radiative heat loss to the surroundings; (2) neglect of incomplete equilibration during the residence time in the combustion chamber; (3) neglect of the enthalpy of vaporization of cryogenic liquid propellants, if they are used; and (4) neglect of incomplete molecular vibrational relaxation due to a less than "perfect" nozzle expansion. The model provides a simple conceptual framework with which to evaluate the potential benefits of hybrid propulsion. It will be shown that by knowing I_{sp} vs. $\Delta E/m$ one can optimize the performance of the propulsion system to suit any set of mission constraints with a correct choice of fuel:oxidizer ratio.

The route to I_{sp} vs. $\Delta E/m$ at a given fuel:oxidizer ratio consists of first calculating the equilibrium temperature and chemical composition of the combustion gases corresponding to a given $\Delta E/m$. Then the I_{sp} can be computed from the expression for the terminal velocity of a "perfect" nozzle expansion. At least two methods that rely on different thermocycles can be used to relate $\Delta E/m$ to the equilibrium combustion temperature and chemical composition. In what might be called a "forward" calculation, one chooses a value for $\Delta E/m$ and energizes the reactants either by an increase in their temperatures or by molecular dissociation at fixed temperature. Then a constant pressure adiabatic flame calculation provides the final equilibrium temperature and chemical composition. In the alternative "reverse" calculation, one first chooses a final temperature and determines corresponding equilibrium chemical composition at a given fuel:oxidizer ratio. Then $\Delta E/m$ is

7

determined from the enthalpy change in going from products at the final temperature to reactants at 298 K. The two methods give equivalent results, but the forward method is used for nearly all of the data reported here because of its greater ease of implementation with the existing Aerospace NEST package of computer programs.[6] Also, as a mathematical convenience, the energizing of the reactants is formally described as a dissociation of molecular hydrogen into H atoms at 298 K.

The coupling of electrical energy to the propellants in an actual thruster is of course a complex process that involves a combination of gas heating, dissociation, and ionization. The present mathematical model does not purport to describe the details of the energy coupling in an electrothermal thruster but rather to give a global description of the thermodynamics of hybrid propulsion based solely on the four assumptions listed at the outset. The calculation of the equilibrium combustion chamber conditions corresponding to a given $\Delta E/m$ is done in terms of state functions (temperature, pressure, and chemical composition) that depend only on the initial and final states of the system; thus, no physical significance is attached to the arbitrary choice of the thermocycle that is used to calculate the final equilibrium conditions that result from a given $\Delta E/m$.

Based on the foregoing, a summary of the more important equations used in this model can now be presented. The electrical energy deposited per unit mass of propellant is given by

$$\Delta E/m = \eta_e \, P_e/\dot{m} = \dot{n}_H \, \Delta H_f/\dot{m} \tag{1}$$

where the term $\eta_e P_e$ is the net electrical power going into the propellants. It is convenient to cast $\Delta E/m$ in terms of α, the initial degree of dissociation of the hydrogen, and ρ, the fuel:oxidizer mole ratio. For the H_2/O_2 case they are defined as

$$\alpha = \dot{n}_H/(\dot{n}_H + 2\dot{n}_{H_2}) \tag{2}$$

8

$$\rho = (\dot{n}_H + 2\dot{n}_{H_2})/(2\dot{n}_{O_2}) \tag{3}$$

from which it is possible to show that

$$\Delta E/m = \frac{\alpha\, \rho\, \Delta H_f}{\rho\, M_H + M_O} \tag{4}$$

Analogous expressions apply to the H_2/F_2 case. For a given value of ρ the range of $\Delta E/m$ is explored by varying α from 0 to 1, which corresponds to an upper limit of $\Delta E/m = 216$ kJ/g in the case of pure H_2 propellant. For those few instances where higher values of $\Delta E/m$ are of interest, the combustion chamber conditions are determined with the "reverse" calculation thermocycle described earlier. The specific impulse is calculated on the basis of an isentropic expansion to infinite Mach number with no further chemical reactions in the nozzle. Thus all of the thermal enthalpy of the combustion chamber gas mixture is converted to exhaust kinetic energy. The average exhaust velocity is given by

$$\bar{v} = g\, I_{sp} = (2\, \Delta H_T/M)^{1/2} \tag{5}$$

where ΔH_T is the enthalpy change in going from combustion chamber gases at temperature T to the same gas mixture at 0 K. The formula for I_{sp} is then given by

$$I_{sp}\,(s) = \frac{(2)^{1/2}(10^3\ \text{J/kJ})^{1/2}}{(9.8067\ \text{m/s}^2)\,(10^{-3}\ \text{kg/g})^{1/2}} \times \left(\frac{\Delta H_T(\text{kJ/mol})}{M\ (\text{g/mol})}\right)^{1/2}$$

$$= 144.21 \left(\frac{\Delta H_T(\text{kJ/mole})}{M\ (\text{g/mol})}\right)^{1/2} \tag{6}$$

The thermal enthalpy ΔH_T and the system average mole weight M are available

from the adiabatic flame calculation. The calculations are run with pure H_2 and with fuel:oxidizer mole ratios of 49:1, 19:1, and 9:1. In terms of the traditional weight ratio of oxidizer to fuel, the data correspond to 0.00, 0.39, 1.00, and 2.11 for H_2/F_2, and to 0.00, 0.33, 0.84, and 1.78 for H_2/O_2. All of the results presented here are for a combustion chamber pressure of 1000 Torr.

III. RESULTS

The calculated I_{sp} curves for the various fuel:oxidizer mole ratios are shown in Figs. 1 and 2 as functions of $\Delta E/m$. These curves give the key results of the analysis and enable one to select the fuel:oxidizer ratio that makes the best use of the available chemical and electrical energy for a given set of mission constraints. The most favorable trade-off between thrust and specific impulse is always achieved by running with the fuel:oxidizer ratio that maximizes I_{sp} at a given $\Delta E/m$. To demonstrate this claim it will help to look at the relationship between thrust and specific impulse,

$$F = \dot{m} \, g \, I_{sp} = \eta_e \, P_e \, g \, I_{sp}/(\Delta E/m) \qquad (7)$$

This equation is used to construct Fig. 3, which shows I_{sp} as a function of the thrust per unit of electrical power, $F/\eta_e P_e$, in the case of H_2/O_2 with the four different fuel:oxidizer ratios. Also shown on the same graph are several lines emanating from the origin for which $\Delta E/m$ is a constant. If, in accordance with the rule stated above, the fuel:oxidizer ratio is chosen so as to maximize I_{sp} for a given $\Delta E/m$, then Eq. (7) and Fig. 3 show that F is also maximized. Thus, for high values of $\Delta E/m$ (> 30 kJ/g) with associated I_{sp} values greater than 850 s, the most favorable trade-off is always achieved with pure H_2 propellant. As I_{sp} decreases it becomes beneficial to add progressively greater quantities of oxidizer until the limiting case of conventional chemical propulsion is reached at an I_{sp} of around 450 s. The combustion chamber temperatures along this optimum curve are in the range of 2000 K to 2700 K for I_{sp} values between 500 s and 950 s; the range extends up to 4600 K at an I_{sp} of 1400 s with pure H_2 propellant. Figures 1 and 2 can be applied to mission modeling by invoking a typical set of constraints and using the curves to derive I_{sp}, $\Delta E/m$, and ρ. Examples of this procedure will be given in the next section. A comparison of Figs. 1 and 2 reveals that H_2/O_2 and H_2/F_2 exhibit very similar performance curves, but that the H_2/O_2 case performs slightly better in the present calculations. In a practical system F_2 is an undesirable choice because of material compatibility problems.
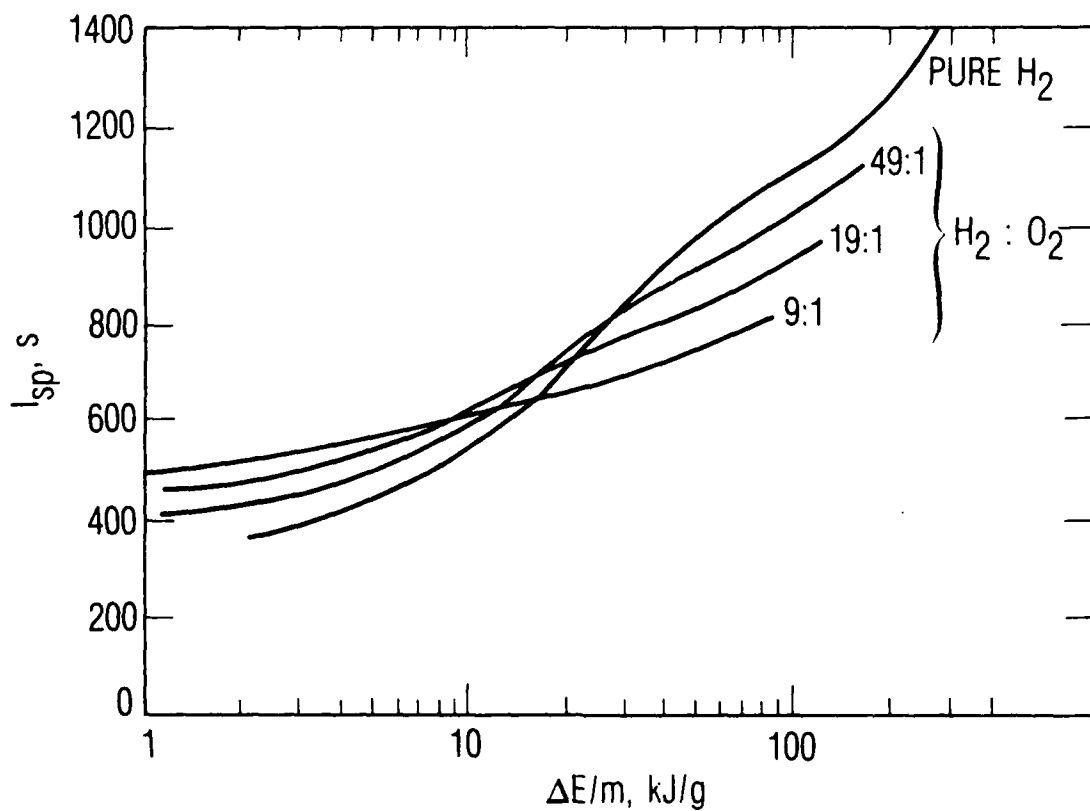
Fig. 1. Specific Impulse vs. Electrical Energy Deposited per Unit Mass
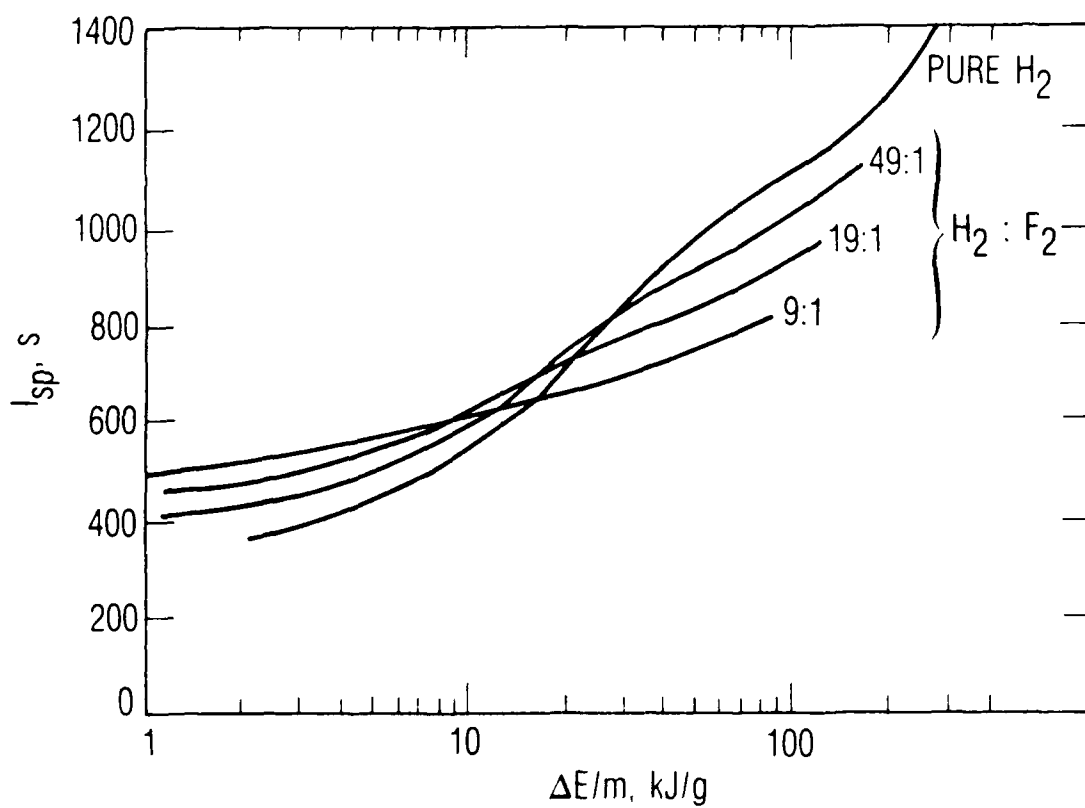of Propellant for H_2/O_2

Fig. 2. Specific Impulse vs. Electrical Energy Deposited per Unit Mass
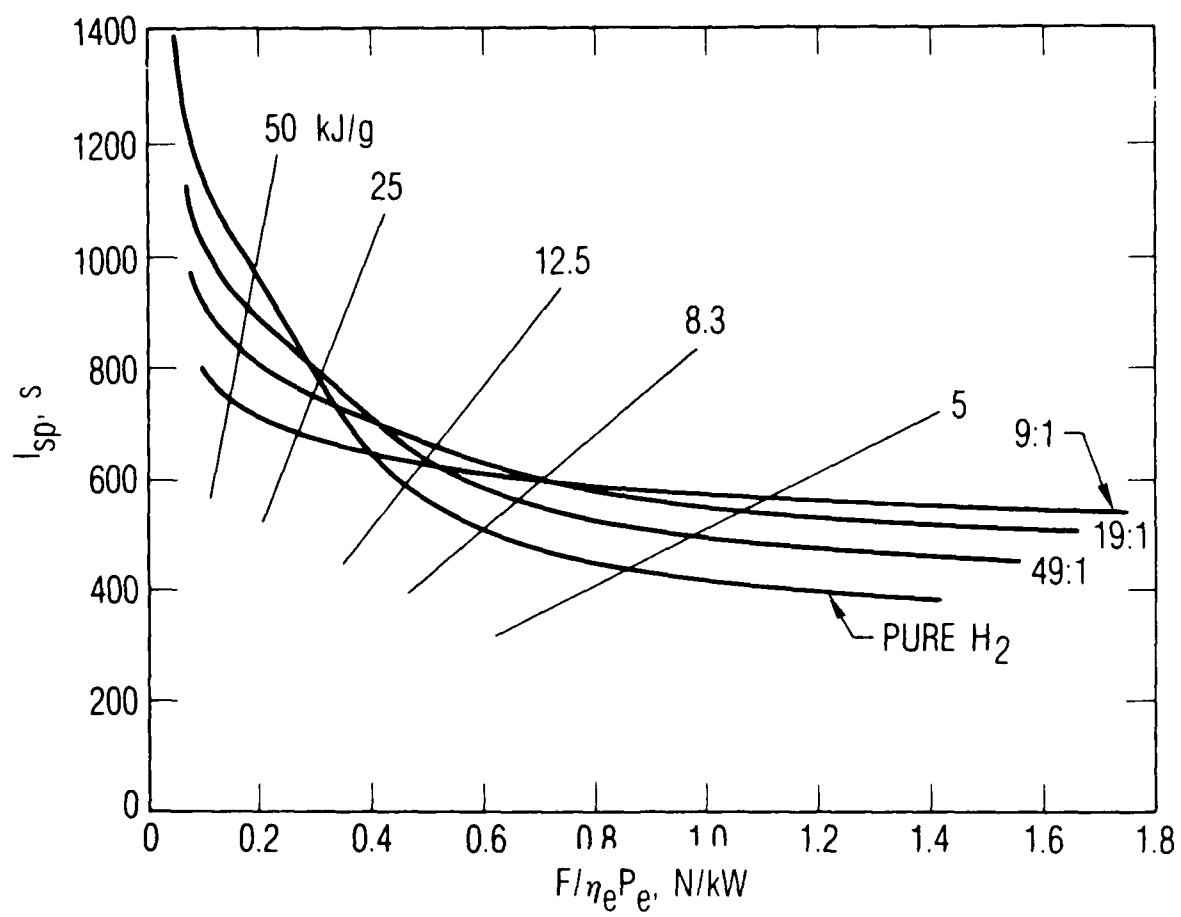of Propellant for H_2/F_2

13

Fig. 3. Specific Impulse vs. Thrust-to-Power Ratio for H_2/O_2

The foregoing results serve to map out the region of applicability of the hybrid electric chemical propulsion scheme based on a conceptually simple but rather idealized thermodynamic model. This model does not account for certain non-ideal modes of behavior that may be exhibited by real electrothermal thrusters. Radiative heat loss to the surroundings can be treated by inserting an appropriate value of η_e, but no attempt is made here to predict η_e for individual thruster designs. A value of η_e in the range of 0.4 to 0.9 is typical for arcjets and resistojets.[3,7,8] Perhaps the most serious limitation of this model is the neglect of chemical kinetic effects that may prevent the attainment of equilibrium during the residence time in the combustion chamber. Although a complete kinetic model of hybrid propulsion is beyond the scope of this paper, a rough estimate can be made of the minimum pressure required for equilibration. The slowest process in the chemical systems we have described is the termolecular recombination of H atoms, $H + H + M \rightarrow H_2 + M$, where M is primarily H or H_2. The rate coefficient[9] is in the range $k_M = 10^9 - 10^{10}$ l^2 mol^{-2} s^{-1} between 300 K and 5000 K. The reaction half-life is given by

$$t_{1/2} \simeq (k_M [H]_0^2)^{-1} \tag{8}$$

where $[H]_0$ is the initial H atom concentration. In the present case, with a pressure of 1000 Torr, the half-life is around 10 μs at 2700 K. This should be quite sufficient to allow equilibration in a resistojet where the residence time is expected to be 100 μs or greater. For an arcjet, however, the residence time could be as short as 10 μs (depending on the electrode geometry), so that pressures somewhat higher than 1000 Torr are required if nonequilibrium effects are to be minimized. In those cases where oxidizer is added to the H/H_2 fuel, the kinetic restrictions imposed by H atom recombination should be relaxed to some extent, because a portion of the available exothermicity is released by fast bimolecular reactions.

The thrust efficiency for hybrid propulsion must be defined carefully to account for the fact that two energy sources (electrical plus chemical) are present. The efficiency for transferring electrical energy into propellant thermal enthalpy η_e has been introduced already. The value of η_e is governed

15

by the mechanism used to energize the propellant gases, be it resistive heating, dc discharge, radio frequency, or microwave. Hence the electrical efficiency η_e is common to all electrothermal propulsion devices. A more interesting quantity from the standpoint of comparing hybrid propulsion with other electrothermal schemes is the gasdynamic efficiency η_{gas}, which is defined here as the efficiency for transferring combustion chamber thermal enthalpy into exhaust kinetic energy. This term is given by

$$\eta_{gas} = \frac{\Delta H_T/M}{(\Delta E/m) + (\Delta H/m)} = \frac{\frac{1}{2} (g\, I_{sp})^2}{(\Delta E/m) + (\Delta H/m)} \qquad (9)$$

where $\Delta H/m$ is the enthalpy change per unit mass of the unreacted propellant gases in going from 0 K to 298.15 K. Thus $\Delta H/m$ is the initial enthalpy per unit mass of the propellants before reaction or electrical excitation. With these definitions one can write down the equation that relates thrust, specific impulse, and electrical power,

$$\frac{1}{2} F\, g\, I_{sp} = \eta_{gas} \left[\eta_e P_e + \dot{m}(\Delta H/m) \right] \qquad (10)$$

In the case of pure hydrogen propellant, η_{gas} is equal to 1 at low I_{sp}, but it drops below unity at high I_{sp} when a significant fraction of the H_2 is dissociated. This is because our assumption of a frozen-flow expansion means that not all of the combustion chamber enthalpy can be converted into exhaust kinetic energy. This effect is shown in Fig. 4, which plots η_{gas} vs. I_{sp} for pure hydrogen electrothermal propulsion and for hybrid electric chemical propulsion. The curve goes through a minimum at around 1400 s and begins to rise again when nearly all of the H_2 has been dissociated and further increments in $\Delta E/m$ go into heating H atoms. Similar plots for H_2 as well as for other pure gases have been published by Jahn.[7] The η_{gas} curve for H_2/O_2 hybrid propulsion is constructed by using Fig. 1 to select the optimum fuel:oxidizer ratio at each I_{sp}. The hybrid η_{gas} curve splits from the pure H_2 curve at around 850 s and rises to much higher values as the amount of available chemical energy increases.
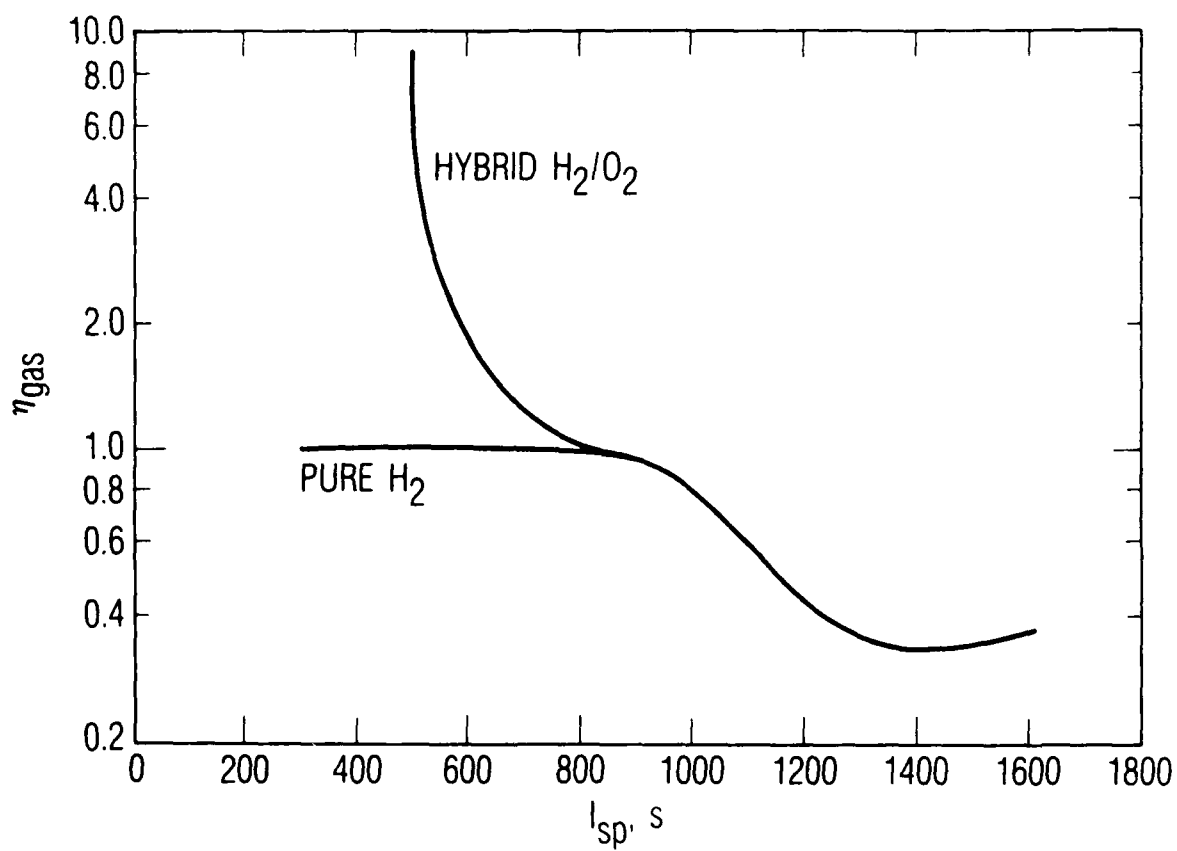
16

Fig. 4. Gasdynamic Efficiency vs. Specific Impulse for Pure H_2 and Hybrid H_2/O_2 Thrusters

IV. APPLICATIONS

The first design example is one in which hybrid electric chemical propulsion is applied to orbital transfer. We consider the case of a LEO → GEO one-way transfer with a final mass of m_f = 10,000 kg and a power level of $\eta_e P_e$ = 50 kW. Subtracting the estimated masses[3] of a nuclear power source, the propulsion system, and the tankage leaves a payload mass in the range m_{pay} = 4000 to 5000 kg. The velocity increment for LEO → GEO with a 28.5 degree inclination change is Δv = 6000 m/s in the low-thrust, continuous-burn case.[3] Figure 5 shows the results of the design calculation for a range of initial masses. The rocket equation

$$I_{sp} = \Delta v / [g \ln(m_i/m_f)] \tag{11}$$

is used to calculate I_{sp}, from which $\Delta E/m$ and the $H_2:O_2$ ratio follow from Fig. 1, and the trip time, Δt, comes from

$$\Delta t = (m_i - m_f)/\dot{m} = (m_i - m_f)(\Delta E/m)/\eta_e P_e \tag{12}$$

Trip time at constant power (50 kW) increases sharply from 76 days at I_{sp} = 850 s to 360 days at I_{sp} = 1500 s, corresponding to the range over which the efficiency η_{gas} for pure H_2 is decreasing. For still higher I_{sp} the trip time actually decreases as the efficiency η_{gas} begins to rise, as is indicated in Figure 4. The hybrid H_2/O_2 curve becomes more favorable than the pure H_2 curve for I_{sp} less than 850 s. Hybrid propulsion offers a significant reduction in trip time (or initial mass) over the pure H_2 case in the range of I_{sp} down to around 500 s, where it converges on H_2/O_2 chemical propulsion. It also adds versatility by being able to tune the performance of the propulsion system over an appreciable range of I_{sp} to match the requirements of the mission. The LEO → GEO transfer was selected for this design calculation because of its central importance for space systems. A power level of 50 kW is probably at the lower limit of the range for which electric propulsion is practical for this particular transfer.
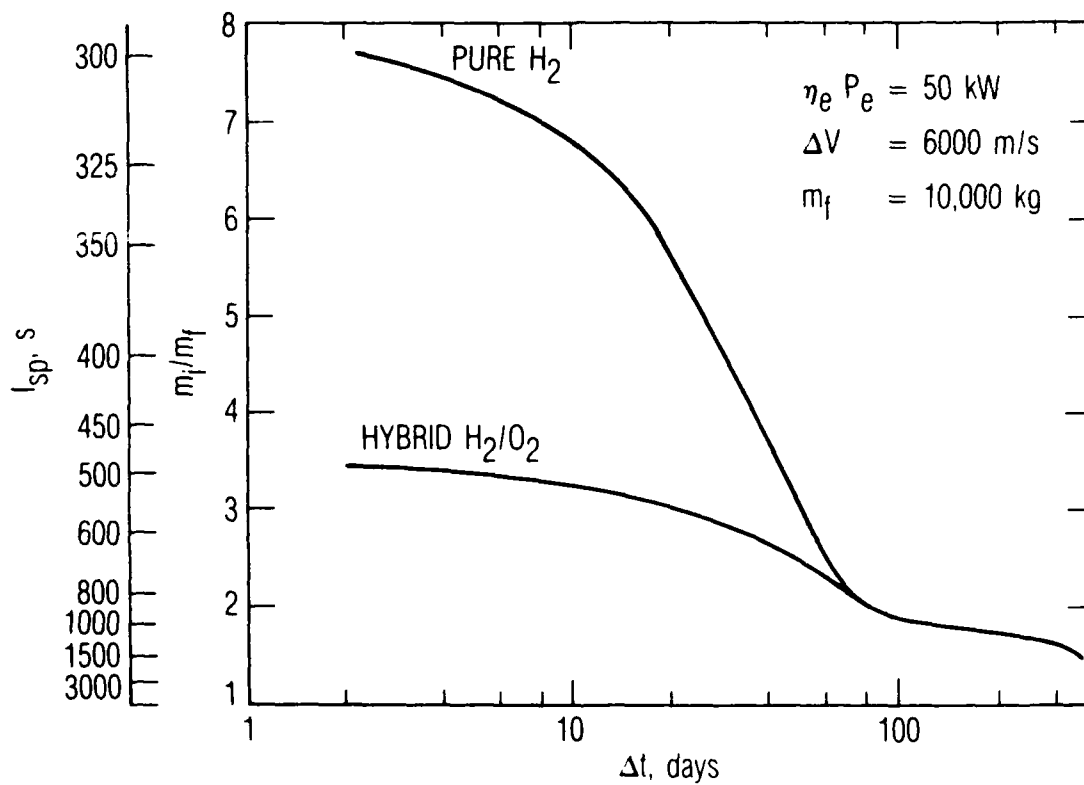
19

Fig. 5. Mass Fraction vs. Trip Time for the Orbital Transfer Thruster
Design Example

Until large space power sources become available, applications for hybrid propulsion will be found in auxiliary propulsion, namely drag make-up, station-keeping, and attitude control. A particularly interesting example would be the case of a manned space station, where the H_2 and O_2 propellants could be derived from an OTV H_2/O_2 propellant tank farm or from electrolysis of H_2O and might be integrated with the life-support system.[1] A design example is given in which hybrid propulsion is applied to a continuous-duty one-newton thruster such as would be used for drag make-up at LEO. Figure 6 shows the trade-off between mass flow rate and electrical power for the pure H_2 and hybrid H_2/O_2 cases. If $\eta_e P_e = 1$ kW then the hybrid thruster uses 15.4 kg/day of propellant, which represents a 28% savings in propellant flow rate over that of the corresponding thruster that runs on pure H_2. If $\eta_e P_e$ is greater than 4 kW, then it is no longer beneficial to add oxidizer. For comparison, a conventional hydrazine monopropellant thruster operating at an I_{sp} of 230 s requires 38.3 kg/day of fuel to perform this mission.
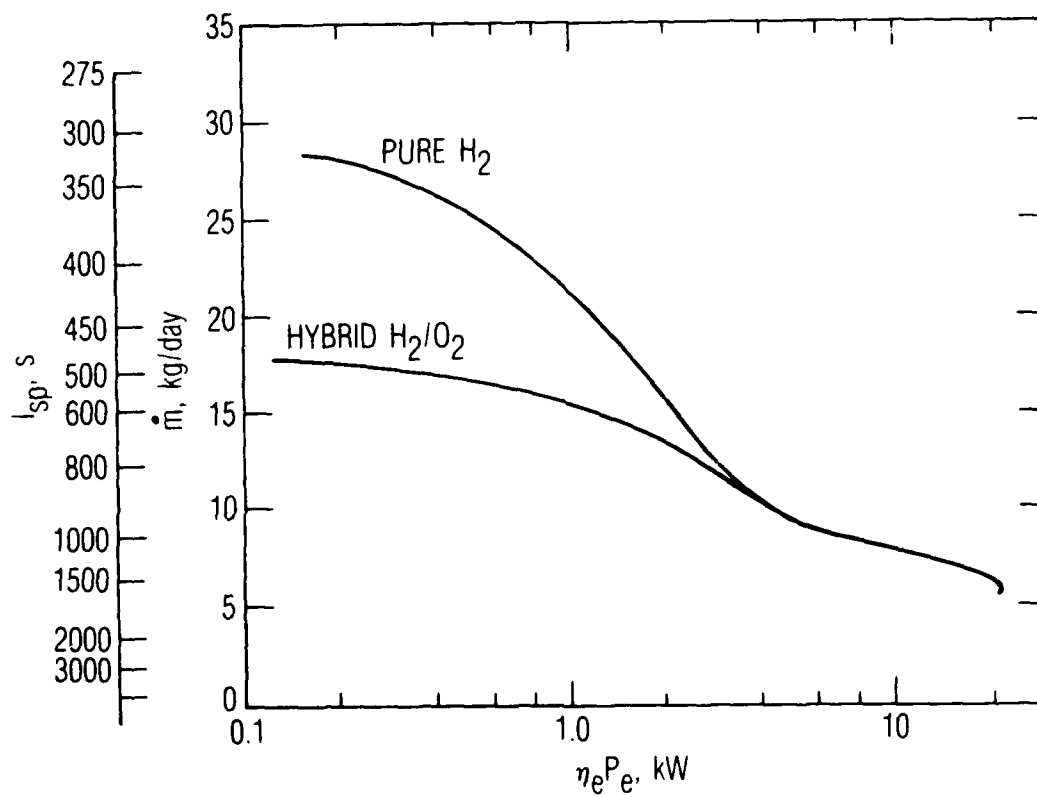
Fig. 6. Mass Flow Rate vs. Electrical Power for the One-Newton Thruster
Design Example

V. CONCLUSION

A hybrid propulsion concept has been described that extends hydrogen electrothermal propulsion by the addition of oxidizer to the propellant flow. Thermodynamic model calculations have been used to predict I_{sp} as a function of the electrical energy deposited per unit mass of propellant. The added degree of freedom provided by the choice of the fuel:oxidizer ratio permits an optimum I_{sp} vs. thrust trade-off to be achieved for any set of operating constraints. Specifically, the I_{sp} and thrust can be simultaneously maximized at any electrical power level and mass flow rate by the correct choice of fuel:oxidizer ratio. Hybrid propulsion represents a significant improvement in performance and versatility in the I_{sp} range up to 850 s when compared with conventional H_2 electrothermal propulsion. Despite the limitations of the model, we feel that the present results show that the hybrid electric chemical concept is potentially a very useful one that deserves further exploration.

REFERENCES

1. R. E. Jones, "Space Station Propulsion: The Advanced Development Program at NASA-Lewis," AIAA Paper 85-1154 (July 1985).

2. R. Poeschel, "Aerospace Highlights 1984: Electric Propulsion," Aerospace America 22, 55 (December 1984).

3. R. M. Jones., "A Comparison of Potential Electric Propulsion Systems for Orbit Transfer," Journ. Spacecraft and Rockets 21, 88-95 (January-February 1984).

4. H. R. Kaufman, "Electric Thruster Performance for Orbit Raising and Maneuvering," Journ. Spacecraft and Rockets 21, 180-186 (March-April 1984).

5. R. H. Frisbee, "A Parametric Study of Thermally Augmented O_2/H_2 Rocket Engines," AIAA Paper 83-1258 (June 1983).

6. K. L. Foster and E. B. Turner, "The NEST Chemistry Computer Program: Operations Handbook," TR-0083(3930-01)-8, The Aerospace Corporation, El Segundo, California (May 1983).

7. R. G. Jahn, Physics of Electric Propulsion (McGraw-Hill, New York, 1968), p. 100.

8. S. J. Sibener, R. J. Buss, C. Y. Ng, and Y. T. Lee, "Development of a Supersonic Atomic Oxygen Nozzle Beam Source," Rev. Sci. Instrum. 51, 167-182 (February 1980).

9. N. Cohen and K. R. Westberg, "Chemical Kinetic Data Sheets for High-Temperature Chemical Reactions," The Aerospace Corporation, El Segundo, California, ATR-82(7888)-3 (July 1982).

NOMENCLATURE

$\Delta E/m$	electrical energy deposited per unit mass of propellant, kJ/g
F	thrust, N
g	9.8067 m/s^2
ΔH_f	standard enthalpy of formation of H atoms at 298.15 K, kJ/mol
ΔH_T	thermal enthalpy of the combustion chamber gas mixture, kJ/mol
$\Delta H/m$	initial propellant enthalpy per unit mass, kJ/g
I_{sp}	specific impulse, s
M	system average mole weight, g/mol
M_H	1.00797 g/mol
M_O	15.9994 g/mol
$\dot{m}$	mass flow rate, g/s
m_f	final vehicle mass, kg
m_i	initial vehicle mass, kg
$\dot{n}_x$	molar flow rate of substance x, mol/s
P_e	electrical power, kW
T	temperature, K
Δt	trip time, s
$\bar{v}$	average exhaust velocity, m/s
Δv	velocity increment, m/s
α	fraction of dissociation
η_e	electrical efficiency
η_{gas}	gasdynamic efficiency
ρ	fuel:oxidizer mole ratio

LABORATORY OPERATIONS

The Aerospace Corporation functions as an "architect-engineer" for national security projects, specializing in advanced military space systems. Providing research support, the corporation's Laboratory Operations conducts experimental and theoretical investigations that focus on the application of scientific and technical advances to such systems. Vital to the success of these investigations is the technical staff's wide-ranging expertise and its ability to stay current with new developments. This expertise is enhanced by a research program aimed at dealing with the many problems associated with rapidly evolving space systems. Contributing their capabilities to the research effort are these individual laboratories:

Aerophysics Laboratory: Launch vehicle and reentry fluid mechanics, heat transfer and flight dynamics; chemical and electric propulsion, propellant chemistry, chemical dynamics, environmental chemistry, trace detection; spacecraft structural mechanics, contamination, thermal and structural control; high temperature thermomechanics, gas kinetics and radiation; cw and pulsed chemical and excimer laser development including chemical kinetics, spectroscopy, optical resonators, beam control, atmospheric propagation, laser effects and countermeasures.

Chemistry and Physics Laboratory: Atmospheric chemical reactions, atmospheric optics, light scattering, state-specific chemical reactions and radiative signatures of missile plumes, sensor out-of-field-of-view rejection, applied laser spectroscopy, laser chemistry, laser optoelectronics, solar cell physics, battery electrochemistry, space vacuum and radiation effects on materials, lubrication and surface phenomena, thermionic emission, photo-sensitive materials and detectors, atomic frequency standards, and environmental chemistry.

Computer Science Laboratory: Program verification, program translation, performance-sensitive system design, distributed architectures for spaceborne computers, fault-tolerant computer systems, artificial intelligence, micro-electronics applications, communication protocols, and computer security.

Electronics Research Laboratory: Microelectronics, solid-state device physics, compound semiconductors, radiation hardening; electro-optics, quantum electronics, solid-state lasers, optical propagation and communications; microwave semiconductor devices, microwave/millimeter wave measurements, diagnostics and radiometry, microwave/millimeter wave thermionic devices; atomic time and frequency standards; antennas, rf systems, electromagnetic propagation phenomena, space communication systems.

Materials Sciences Laboratory: Development of new materials: metals, alloys, ceramics, polymers and their composites, and new forms of carbon; non-destructive evaluation, component failure analysis and reliability; fracture mechanics and stress corrosion; analysis and evaluation of materials at cryogenic and elevated temperatures as well as in space and enemy-induced environments.

Space Sciences Laboratory: Magnetospheric, auroral and cosmic ray physics, wave-particle interactions, magnetospheric plasma waves; atmospheric and ionospheric physics, density and composition of the upper atmosphere, remote sensing using atmospheric radiation; solar physics, infrared astronomy, infrared signature analysis; effects of solar activity, magnetic storms and nuclear explosions on the earth's atmosphere, ionosphere and magnetosphere; effects of electromagnetic and particulate radiations on space systems; space instrumentation.

END

FILMED

6-89

DTIC